Spacecraft Charging, An Update

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***Abstract*—Twenty years after the landmark SCATHA program, spacecraft charging and its associated effects continue to be major issues for earth-orbiting spacecraft. Since the time of SCATHA, spacecraft charging investigations were focused primarily on surface effects and spacecraft external surface design issues. Today, however, a significant proportion of spacecraft anomalies are believed to be caused by internal charging effects (charging and ESD events internal to the spacecraft Faraday cage envelope). This review will, following a brief summary of the state of the art in surface charging, concentrate on the problems introduced by penetrating electrons ("internal charging") and related processes (buried charge and deep dielectric charging). With the advent of tethered spacecraft and the deployment of the International Space Station, low altitude charging has taken on a new significance as well. These and issues tied to the dense, low altitude plasma environment and the auroral zone will also be briefly reviewed.**

***Index Terms*—ESD, space plasma interactions, space weather, spacecraft charging.**

I. Introduction

THE growing sophistication of spacecraft has lead to increasing concern over spacecraft environment interactions associated with plasmas. Over the last two decades there have been numerous conferences [1]–[4] and books [5]–[7] on the generic issues associated with plasma interactions or specific areas such as surface charging. Still, as Koons *et al.* [8] recently demonstrated, charging (or perhaps more properly differential charging followed by discharging) effects are still a major source of spacecraft anomalies (see also [9] and [10]). Whether it be surface charging, internal charging, plasma interactions at low altitudes, or induced fields on tethers, the buildup of charge on or in spacecraft poses a continuing problem for the spacecraft builder.

Garrett [11] reviewed the field of spacecraft surface charging as of 1980. Spacecraft charging, defined in that review as the buildup of charge on spacecraft surfaces, has been of concern to users and operators of spacecraft since the first days of the space age. In that original review, the study of spacecraft charging was characterized by four phases. The first phase, the "pre-space age," was primarily concerned with the theory of simple probe charging and with rocket measurements of charging in the ionosphere. It ended in 1957 with the launch of Sputnik. The brief second phase was marked by the formal foundations of charging theory (at least in the ionosphere) and by the first tentative measurements by satellites. The third phase, in the early 1960s, was characterized by the first accurate measurements of charging on spacecraft and rockets. Self-consistent charging models were developed and factors such as secondary emission and photoelectron currents were included in these models. It ended roughly in 1965 with the publishing of Whipple's thesis [12] on spacecraft surface charging. That thesis and reviews by Brundin [13], Bourdeau [14], and others established the basic components of charging theory and the range of observations. The fourth phase, 1965–1980, was characterized by increasingly more sophisticated models of spacecraft surface charging, *in situ* measurements, and definition of the space plasma environment. Giving impetus to the study of spacecraft charging, the first *in situ* observations of kilovolt potentials at geosynchronous orbit were reported by DeForest [15], [16] in 1971. This period ended with the flight and analysis of the SCATHA (P78-2) spacecraft. Reviews by Garrett [11] and Whipple [17] summarized the major theoretical and observational findings of the period. The engineering implications of these findings were summarized in the NASA Spacecraft Charging Design Guidelines [18] and MIL STD 1541A [19]. Thus ended the first 20 years of spacecraft charging studies.

This review will provide an overview of the changes in the field of spacecraft charging between 1980 and the present, the next 20 years. The nearly 20 years between the original review in 1980 and now mark a fifth "age of charging." The changes since 1980 have in general been in emphasis as there has been a major shift in attitude *vis-a-vis* surface charging versus internal charging caused by penetrating electrons. While the former continues to be an important process, in recent years it has become increasingly clear that, as external charging and the elimination of differential potentials are routinely addressed in spacecraft design, a growing proportion of spacecraft anomalies are now believed to be caused by "internal" charging (defined as charging not on the external visible surface of the spacecraft, but by charging that causes discharges near internal electronics). To address this issue, a new NASA Handbook, "Avoiding Problems Caused by Spacecraft On-Orbit Internal Charging Effects" [20] has been written. Likewise, with the importance of the International Space Station to the national space program, charging effects unique to the low earth orbit have become of increasing concern. Finally, the continuing desire to use high voltages in space (especially for solar arrays) and to utilize tethers have in particular led to growth in these areas during the fifth period.

II. Surface Charging

Surface charging in this paper refers to charging effects and electrostatic discharge effects on the outside of the spacecraft (generally the visible surface materials). It is now universally

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recognized as an important design consideration for spacecraft. Surface charging is defined by the current balance equation

$$I_T(V) = -I_E(V) + (I_I(V) + I_{\mathrm{SE}}(V) + I_{\mathrm{SI}}(V) + I_{\mathrm{BSE}}(V) + I_{\mathrm{PH}}(V)) \quad (1)$$

where:

- V surface potential relative to space;
- I_T total current to spacecraft surface at V; $=0$ at equilibrium when all the current sources balance;
- I_E incident negative electron current;
- I_I incident positive ion current;
- I_{SE} secondary emitted electron current due to I_E;
- I_{SI} secondary emitted electron current due to I_I;
- I_{BSE} back scattered electron current due to I_E;
- I_{PH} photoelectron current.

The solution of (1) can be quite complicated [11], [12], [17]. Subject to various constraints (e.g., Poisson's equation and the time-independent collisionless Boltzmann or Vlasov equation), it is the fundamental relationship for determining surface potentials. Briefly, each of the current terms on the right-hand side of (1) are determined as a function of potential to give so-called I–V curves. The equation is then solved (subject to the aforementioned constraints) so that $I_T(V) = 0$. Currently, a common procedure for geosynchronous orbit is to approximate the ambient environment in terms of Maxwellian or two Maxwellian plasma distributions. Then, dependent on the geometry, I–V curves for the electrons and ions can be readily estimated by simple analytic expressions. As material secondary emission properties have been shown to have a strong influence on surface charging [11], [21]–[23], the secondary, back scatter, and photoelectron current terms typically have to be included if quantitative estimates of the spacecraft potential are required (other current terms such as for artificial plasma beams may also be included in (1) but will not be discussed here). One complication is the so-called "triple root." First recognized by Whipple [12] and expanded on by subsequent authors, (1) can have multiple roots and in principle the solution can jump between these "triple-roots" [24]–[28], perhaps being the source of the sudden high voltage jumps in the surface potential which could cause arcing. In any event, (1) has been solved to give the spacecraft potential under a variety of conditions.

An example of a first-order solution of (1) for the earth's magnetosphere is presented in Fig. 1 [29]. Fig. 1 is an approximation of the expected range of the charging threat in terms of surface potential as a function of altitude and latitude in the absence of photoemission. It is a worst case estimate of charging for a conducting spherical aluminum satellite at a given altitude and latitude. This figure is intended to serve as a simple tool for mission planning. If a spacecraft's orbit passes through one of the high potential regions, a project should either take steps to mitigate surface charging or do an analysis to assess the risk to the spacecraft from differential charging. (The authors have found that Fig. 1, while perhaps not strictly quantitatively accurate, provides a powerful means of alerting project managers to the need to address spacecraft charging in the satellite's design.) The primary region of surface charging is, as has been recognized for many years, in and near geosynchronous orbit and along the

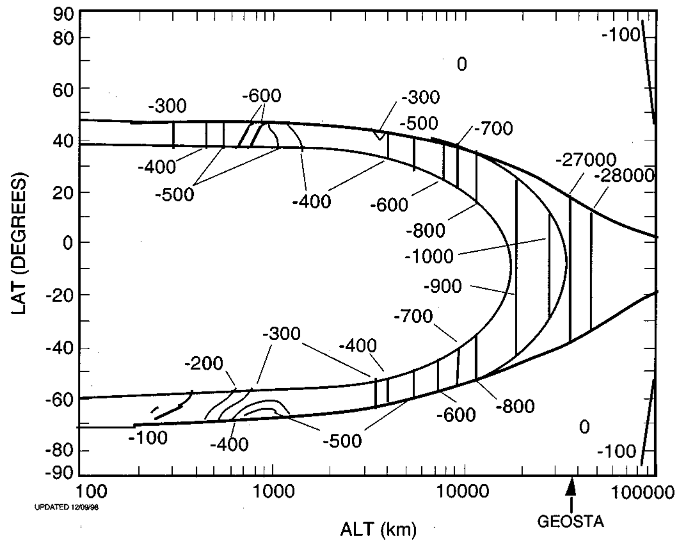

Fig. 1. Surface charging potential contours (in the absence of sunlight) as a function of altitude and latitude for a conducting spherical aluminum satellite [29].

field lines extending down into the auroral zones. This region was extensively mapped by the SCATHA satellite and the characteristics of the environment presented in a series of descriptive "atlases" [30]–[36] and worst case studies [18], [37]–[41]. Of increasing interest, however, is the portion of the charging environment below 1000 km in the polar regions. Although not as dramatic as geosynchronous charging, "low altitude" surface charging in this region is more common than originally thought (see Section IV and review by Hastings [42]).

The spacecraft surface charging environment has been mapped out for other planets; surface potentials have been estimated for Jupiter and Saturn [43], [44]. In support of such estimates, the Voyager spacecraft may have observed large surface charging throughout the solar system, possibly tens of kilovolts at Jupiter [45] and −400 V at Uranus [46]. Many interplanetary spacecraft are now, as a result, designed to minimize surface charging as a matter of course. These design techniques are based on design guidelines and standards defined in NASA 2361 [18] and MIL-STD 1541A [19]. The methods for controlling and mitigating surface charging were the direct outgrowth of the SCATHA experience [47]–[49]. Actual flight experience over the last decade has repeatedly demonstrated the value of these methods. Indeed they have consistently proven to be successful in limiting the effects of surface charging.

Although it is still difficult to adequately predict geomagnetic "weather" in terms of substorms with anything more than a half to one hour lead time [50]–[53], it has proven possible to estimate absolute surface charging levels at a given satellite location with some accuracy from geomagnetic indices [54], [55] or, better still, *in-situ* measurements of the plasma (note: differential charging is another matter altogether and requires intimate knowledge of the spacecraft design and sophisticated codes such as NASCAP [56] to provide accurate estimates). In Garrett *et al.* [57], data from plasma sensors on one geosynchronous spacecraft were successfully used to estimate charging levels at an-

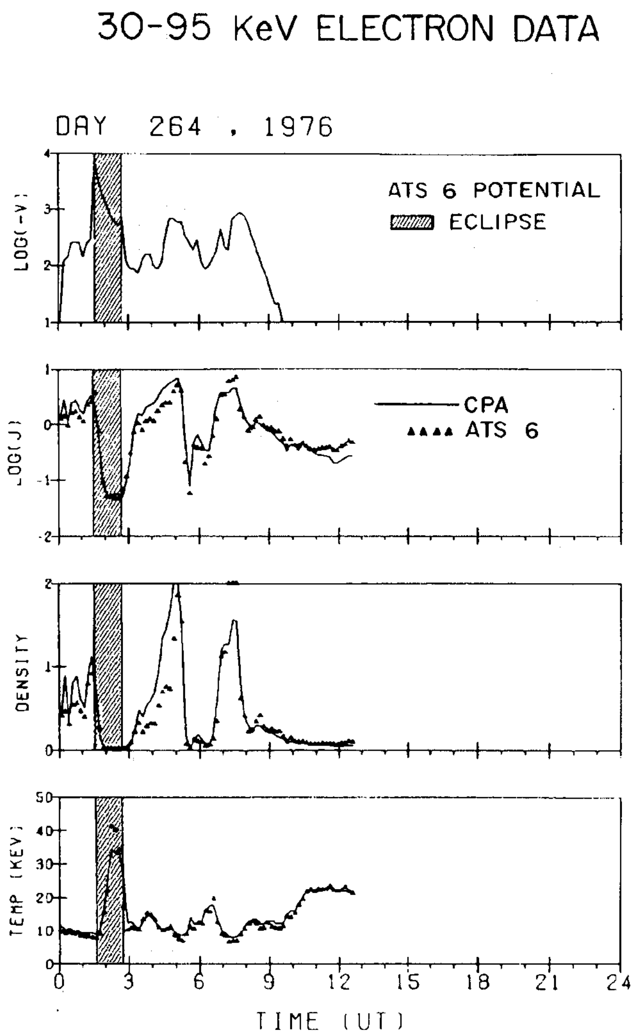

Fig. 2. Measurements of the 30–95 keV electron channels from the DSP CPA instrument compared with the plasma and charging environment at ATS 6 [57]. As demonstrated in the reference, the DSP CPA 30–95 keV electron current (second frame from the top) was shown to be a proxy for the ATS 6 surface potential variations when the different relationships between the current and potential in eclipse and in sunlight were accounted for. This relationship held up even when the spacecraft were separated by several earth radii.

other spacecraft. These measurements, obtainable in near-real time, can be used to estimate charging levels at other spacecraft within several hours of local time around the observing spacecraft. This capability is demonstrated in Fig. 2 [57] where data from one spacecraft (the USAF Defense Support Program or DSP satellite) were used to estimate the charging environment on the near-by ATS-6. Of interest is that this was done with only three electron energy channels between 30 and 95 keV. The results also demonstrated that surface charging is primarily a function of the electron current at energies of a few tens of kilovolts and that it is possible to provide a "spacecraft surface charging index."

Despite these advances, surface charging at geosynchronous orbit can still pose a threat to spacecraft survivability [8], [58]–[60]. Recently, new evidence for the complexities associated with the surface charging/arcing process has emerged in the form of catastrophic, continuous arcs between adjacent solar cells on two high-powered spacecraft operating in geosynchronous orbit [61]. Ground experiments and theory have shown that the most probable location for electrical discharges to occur on the surfaces of high voltage solar arrays is at the so-called triple junction: the interface between a metallic interconnect, coverglass, and plasma [62]–[64]. Although this type of arc is not believed to be able to cause substantial damage to a solar array, it has been hypothesized and demonstrated in the laboratory that such an arc can generate sufficient local heating to initiate outgassing and polymer pyrolysis [64], [65]. This in turn can generate enough gas and plasma between biased solar cells to trigger long duration arcs that can be maintained by the solar array and cause serious damage to an array. Fortunately, mitigation techniques [64], [65] (e.g., limiting the potential between adjacent solar cells and insulating the region between likely breakdown sites) have proven in testing to be very effective at reducing this problem.

III. Internal Charging

Internal charging as used here refers to the accumulation of electrical charge on interior ungrounded metals or on or in dielectrics inside a spacecraft. The key difference between "internal" and external/surface charging is that surface electrostatic discharges often are loosely coupled to victim circuits, whereas internal discharges may occur directly adjacent to victim circuits. If Faraday cage construction is employed, ESD events outside the Faraday cage, even if under thermal blankets, could be called "external" in this context. Fig. 3 shows electron and proton ranges in aluminum versus energy. Since most satellites have an outer shell with aluminum equivalent thickness of 30 or more mils (0.76 mm), internally deposited electrons usually had external energies greater than 500 keV. Thus electrons with 500 keV of energy or more are considered to be the primary environment responsible for internal charging problems. Although the fluxes are lower at these higher energies, any internal electrostatic discharge (ESD) spark they might cause is closer to victim electronics than external ESDs and therefore can cause significant upset or damage to satellite electronics. Note, however, that "internal" charging as defined herein can occur under thinner protective layers (as thin as a thermal blanket) and the energy threshold for internal charging can be caused by electron environments of energies perhaps as low as 100 keV. The key is the deposition rate/fluence after shielding effects have been taken into account (as described later in this section).

During the Voyager 1 passage by Jupiter on September 5, 1977 [66], [67], 42 identical electrical anomalies were observed. These were subsequently attributed to internal charging. In particular, it was postulated that $\sim$MeV electrons had penetrated the surface of a cable and built up charge sufficient to cause arcing. Analysis of SCATHA, CRRES, and DSP data [68] showed similar effects. Laboratory studies by Leung [69], Frederickson [70]–[72], and others demonstrated that internal (also called buried or deep dielectric) charging was a potential source of the discharges. As a result, a series of internal charging experiments were flown on the CRRES spacecraft in 1990–1991 [72]. These experiments, which exposed a variety of configurations of isolated conducting surfaces and dielectrics to the earth's radiation environment,

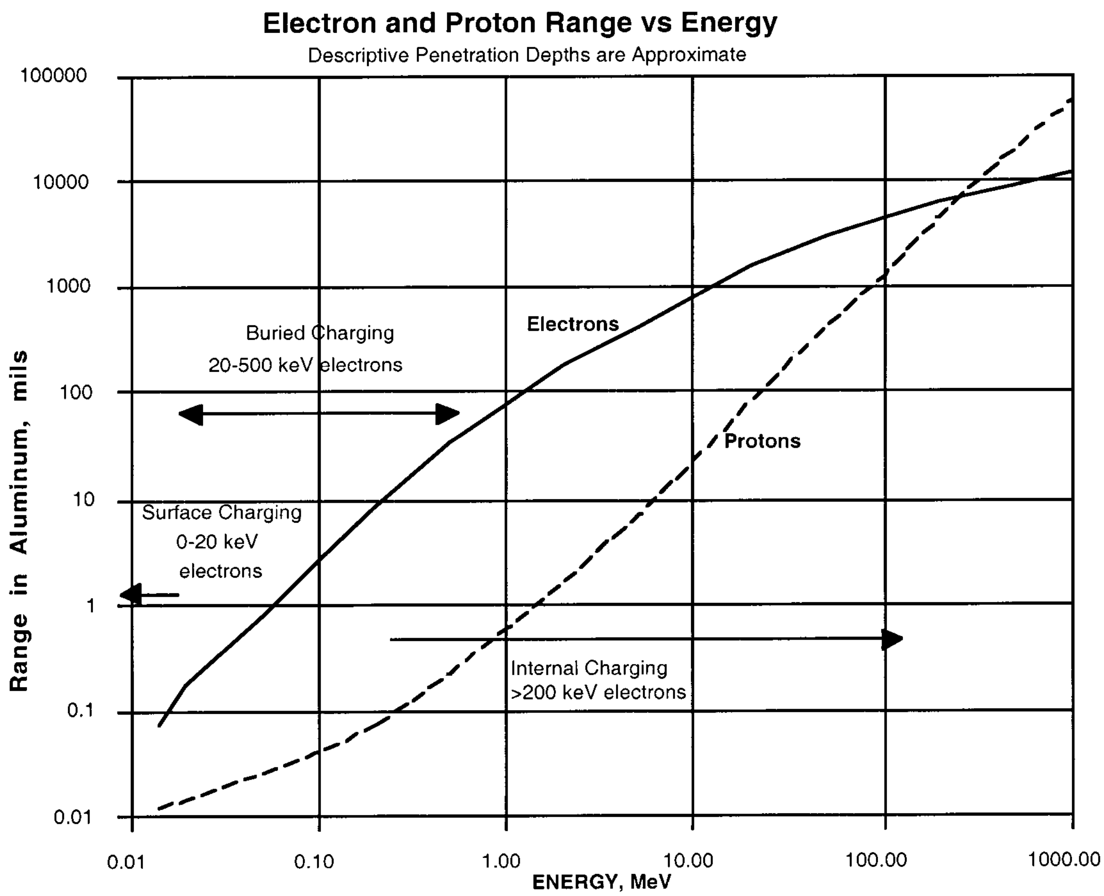

Fig. 3. Electron and ion penetration ranges in aluminum [20]. (1 mil = 0.0254 mm.)

clearly demonstrated the reality of this effect. Over 4000 pulses were detected during the 13 month lifetime of the CRRES spacecraft. As in the case of SCATHA for surface charging, CRRES marked a watershed in the study of internal charging. Since that time, the presence of internal charging continues to be investigated and reported [9], [58], [73]–[83].

The basic high energy electron environment model is still NASA AE8 [84], [85]. Various enhancements have been proposed to refine it [78], [79], but they are not in common usage as yet. The hazard to earth orbiting spacecraft arises from the high energy electron environment that peaks at about 1.5 R_e and 4 R_e equatorial. This is shown alternatively in Fig. 4 as a parallel to Fig. 1. As in Fig. 1, this figure is to be used as a screening tool for internal charging problems. The sub-GEO orbits carry the greatest risk, but the GEO orbits have exhibited more problems because this is a more populated region.

The insulators that are the greatest threat to internal electronics are those that are closest to victims. Circuit board dielectrics with accumulated charges can be immediately adjacent to a victim integrated circuit and can couple nearly the full stored energy if an ESD is unfavorably directed. NASA-HDBK-4002 [20] recommends no more than 3 cm^2 of open circuit board area. The circuit board threat model assumes a typical FR4 board material about 80 mils (2 mm) thick and with no traces or metallization in that region. If traces or ground planes are located in that region, even if not visible on the surface of the board, the leakage paths increase and the threat level diminishes. There is ongoing work to quantify the effects of embedded ground planes on discharges, but actual design rules are in their infancy.

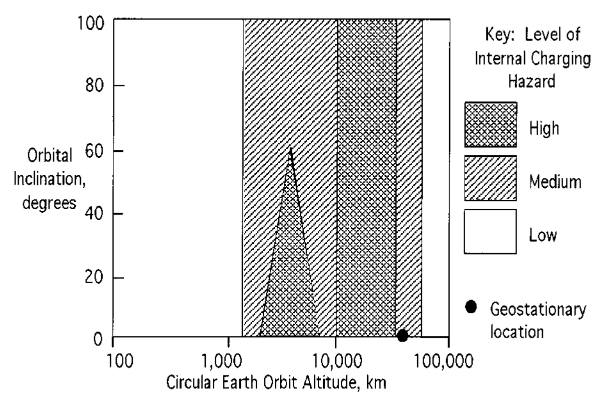

Fig. 4. Internal charging threat for circular earth orbits [20].

Cable insulation is another dielectric that poses an internal charging threat. There is research and published literature about the effects of cable insulation as an internal charging threat [74], but as yet there are no specific quantitative guidelines. The threat starts with charging in the dielectric (so that thinner wire insulation is better) and the threat is enhanced by small plasma plumes drifting onto victim areas (such as exposed high voltage terminals), so keeping the wire bundle tightly bound is better. Otherwise, research is ongoing at this time.

Radiation-induced conductivity is a real phenomenon describing temporary or permanently enhanced conductivity due to ionizing radiation. It tends to mitigate internal charging, but the quantitative aspects of it have not been published in a usable form (if they are known at all). Therefore, threat assessments and spacecraft design requirements are best done without assuming the benefits of radiation-induced conductivity. An effect that worsens internal charging is that dielectrics in space seem to lose some of their earth-measured conductivity. Most arcing or pulsing seems to take place some time after launch rather than immediately upon arriving on station. Part of this phenomenon can be attributed to drying of the material in the vacuum of space creating a greater surface and/or bulk resistivity and a resultant enhanced rate of ESD events in that dielectric.

The problem of internal charging will be getting worse in the future rather than better. Although the environment is not changing, economics is forcing less and less shielding, lighter weight structural materials, and even elimination of Faraday cage construction. At the same time, the devices (integrated circuits) are going to smaller scale sizes and thus are more susceptible to damage. The designer has an even greater challenge ahead; design rules that may have worked yesterday will need re-evaluation with changed parts and structural design.

Except for bulk conducting materials, charge will be deposited over a finite depth; indeed, any particle with energy over a few eV will penetrate the surface. The depth of penetration and charge deposition is a function of stopping power, the energy of the impinging particles, and any electric fields normal to the surface (see Fig. 3 for the penetration depth of energetic electrons and protons in aluminum). A common spacecraft surface configuration that will exhibit this behavior consists of an exposed dielectric material with a conductive backing connected to the spacecraft ground. Charge will accumulate (or diffuse away) in the dielectric over time as a function of the conductivity of the material and the imposed electric fields. If the charge accumulating in the dielectric induces a field greater than the breakdown strength of the material (typically of the order of 10^5 to 10^6 V/cm), a discharge can occur within the material or from the interior of the dielectric to one of its surfaces. This occurs with an electron charge deposition on the order of 2×10^{10} electrons/cm^2.

The computation of internal charging resembles surface charging calculations with the inclusion of space charge. The basic problem is the calculation of the electric field and charge density in a self-consistent fashion over the three-dimensional space of interest. The primary difference between the two is the role that the conductivity of the material plays in the process. Poisson's equation must be solved subject to the continuity equation in the dielectric. As a very simple example, consider a one-dimensional, planar approximation at a depth X in the dielectric. The equation at X is then:

$$\varepsilon(dE/dt) + \sigma E = J \tag{2}$$

where E is the electric field at X, t is time, σ is the conductivity in (ohm-m)$^{-1}$ (= $\sigma_o + \sigma_r$). Here σ_o is the dark conductivity, σ_r is the radiation induced conductivity [86], ε is the dielectric constant. J is the incident particle flux (current density) at X including primary and secondary particles. A solution of this equation for σ and J independent of time is

$$E = E_o \exp(-\sigma t/\varepsilon) + (J/\sigma)(1 - \exp(-\sigma t/\varepsilon)) \tag{3}$$

where E_o is the imposed electric field at t = 0.

Although only a crude approximation, these two equations demonstrate the basic features of radiation induced charging. In particular, they demonstrate the importance of the charging time constant τ (= ε/σ). For many materials, τ ranges from 10 to 10^3 s. Some common dielectric materials used in satellites have even longer constants of 3×10^5 s. In regions where the dose rate is high (enhancing the radiation conductivity), the E field comes to equilibrium rapidly. In lightly irradiated regions, where the time constant is long (the dark conductivity dominates), the field takes a long time to reach equilibrium. Depending on the dielectric constant and resistivity, as a rule of thumb, 10^{10} to 10^{11} electrons/cm^2 on the interior of a spacecraft may cause internal discharges. Electron energies of importance are between 100 keV to 3 MeV for typical spacecraft construction and most earth orbits. Charging times at these energies and the fluxes at geosynchronous orbit would be about 3–10 h. At lower charging rates, material conductivity often leaks off the charge so that internal charging would not be a problem. In Fig. 5, measurements of the $E > 1.2$ MeV electrons at geosynchronous orbit by the GOES-2 satellite between July 1980 and May 1982 and star-sensor anomalies (vertical arrows) on the DSP satellite are seen to be well correlated, confirming this proposition [68].

As in the case of surface charging, there are currently on station several geosynchronous monitors that can be used to provide a real-time "internal charging index." When the high energy electron flux exceeds a critical number for a given spacecraft (e.g., 1000 flux units in Fig. 5), arcing may occur.

Although many of the procedures for limiting surface charging can be applied to internal charging, there are issues specific to internal charging that are not covered in NASA TP-2631 [18] or in MIL-STD 1541A [19]. Until recently, there was not a consensus in the spacecraft engineering community as to what and to what degree design features are necessary to limit internal charging effects. This lack of consensus resulted in several spacecraft suffering upsets that might have been avoided if proper guidelines had been in place. As an example, on January 20 and 21, 1994, the Anik-E1 and E2 spacecraft suffered serious upsets within hours of each other that resulted in the brief loss of one and a six month outage of the other. Subsequent analysis has implicated internal charging as the cause [87], [88].

Knowledge of charge in internal dielectrics (and/or ungrounded conductors) is key to quantitatively evaluating the internal charging threat. The same radiation transport codes used for radiation dose calculations have the necessary information (geometry, thicknesses, material types, etc.) to calculate charge deposition, but most of them do not report the deposited charge. One approximate conversion factor to determine charge deposition is that the number of electrons/cm^2 equals 2.4 $\times$ 10^7 $\times$ dose (rads Si) [89]. A code called EGS4 has been used with some success to evaluate internal charging levels

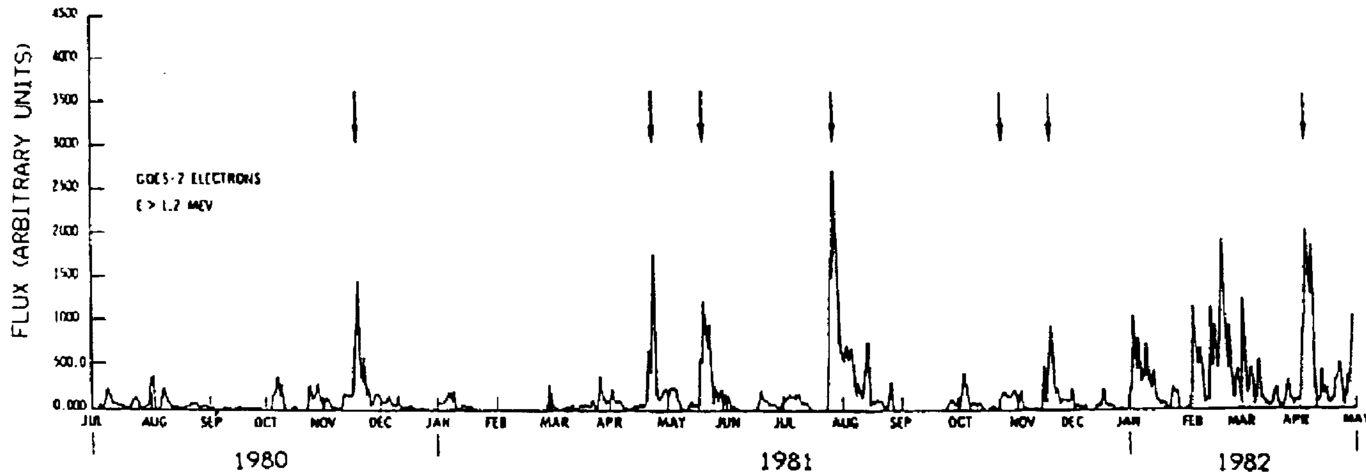

Fig. 5. GOES-2 $E > 1.2$ MeV electron flux at geosynchronous orbit between July 1980 and May 1982 compared with star-sensor anomalies (indicated by vertical arrows) on the DSP satellite [68]. (Reprinted by permission of Elsevier Science Publishers.)

[82]. Other codes are described in NASA-HDBK-4002 [20]. NASA-HDBK-4002 [20] also contains a good overview of the subject of internal charging and contains most of the knowledge from these and similar studies. It defines the primary earth orbits where internal charging might be of concern. A tutorial is provided on the internal charging process and a checklist for designers to use in preventing internal charging (e.g., a basic set of design rules).

IV. Low Altitude Charging

Spacecraft orbiting at low altitudes must also be concerned with charging. Because of the complex effects of structure size and shape on the magnetohydrodynamic flow fields of high density plasmas, hypersonic plasma interactions at low altitudes have always presented an analytic challenge [90]. Likewise, the desire to operate at increasingly higher solar array voltages have greatly added to the computational difficulties associated with this problem. Fortunately, with the continuing growth in low cost computing capability, a number of spacecraft charging problems at low altitudes are for the first time yielding to numerical analysis. Intricate geometries, magnetic fields, changing composition, and high, imposed potentials can now all be effectively modeled. As outlined in Hastings' review [42], low altitude charging analysis is coming of age.

Turning to the basic physics, the low altitude charging problem is best represented by the movement of a body through a dense, cool ionospheric plasma. For a typical spacecraft, its characteristic dimensions are, in contrast to geosynchronous orbit, quite large compared to the plasma debye length. This factor makes current flow computations for complex geometries and field configurations difficult. The basic variations, however, are related to first order to neutral gas flow around a body. These variations can be easily illustrated in the case of the current flow to a small spacecraft in low earth orbit. In Fig. 6 predictions for a simple cylindrical geometry [91], as a function of altitude (and hence composition), are compared with actual data from measurements on a small spacecraft [92]. This figure demonstrates how current flow varies dramatically with angle and how the depth of the wake varies with altitude/composition. Similar measurements for a large body (e.g., the Shuttle) have also been made [93]–[96].

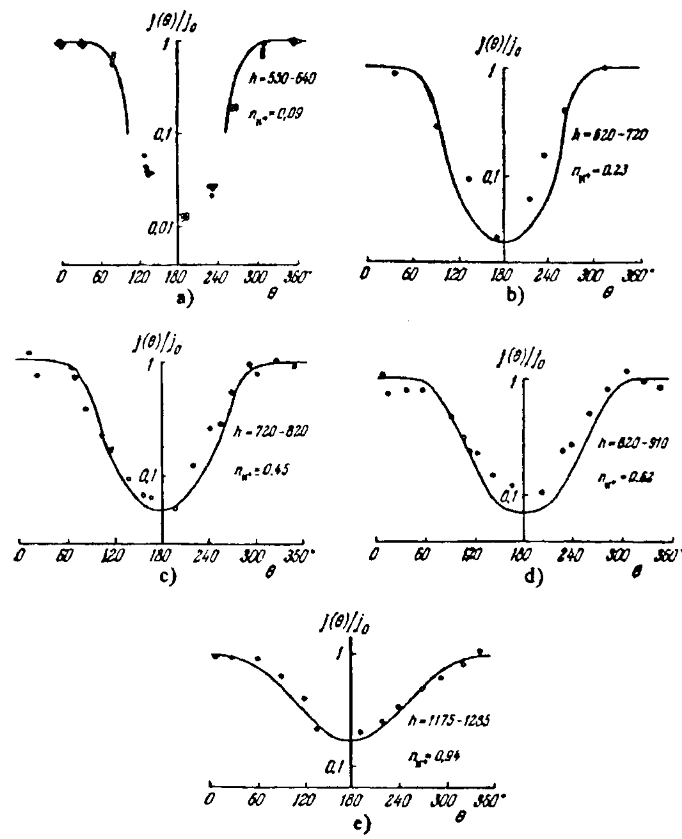

Fig. 6. Normalized electron current $(j(\theta)/jo)$ versus angular position of the plasma probe on Explorer 31 [92]. 180° would correspond to the center of the wake (i.e., opposite the direction of movement). The altitude h and ambient density n are indicated. (Reprinted by permission of the American Institute of Physics.)

When magnetic fields, imposed potentials, and complex geometries are also included, the difficulty of the problem departs dramatically from that for a simple neutral gas flow and can

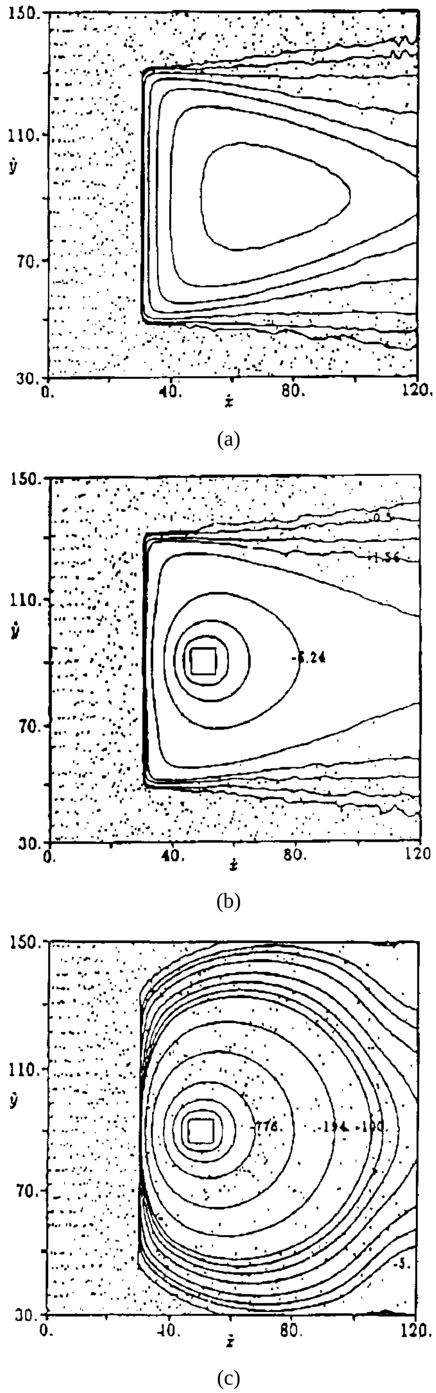

Fig. 7. Low altitude potential and ion flow contours for four different conditions (see text) [97].

seldom be addressed analytically. As a sample of the difficulties that a typical problem can introduce, consider a large biased plate in low earth orbit. Fig. 7 is a plot of the plasma flow field for a large flat plate (representing say a large solar array panel) at low altitudes. This figure [97] illustrates several possible variations. The first frame, Fig. 7(a), is for an unbiased plate in a low altitude ionospheric plasma at Mach 8. Next, a small, isolated body is inserted in the flow field behind the plate and allowed to float to an equilibrium potential. This alters the flow field slightly [Fig. 7(b)]. Next, an externally imposed current source is applied (an auroral particle beam such as observed at high inclinations). The main plate does not alter its potential significantly but the smaller body, because it is shielded from the ionospheric plasma, begins to charge. As it does so, it significantly alters the wake flow [Fig. 7(c)]. In Fig. 7(d), a potential difference is applied between the plate and the small body (this might correspond to a crew module biased relative to the main arrays for the International Space Station). As the potential difference is increased, the flow field becomes even more altered.

Experimental work related to this phenomenon (i.e., current collection by high voltages in a wake) have been carried out in the laboratory [98] and *in-situ* by the Shuttle Charging Hazards and Wake Studies (CHAWS) experiment [99], [100]. CHAWS consisted of plasma monitors and a biasable probe mounted on the Shuttle Wake Shield Facility (WSF). The experiment measured the plasma current in the wake of the WSF as a function of the negative potential of the probe (up to -5000 V relative to the WSF). The experiment was modeled using the programs Potentials of Large Objects in the Auroral Region (POLAR) [101], [102] and Dynamic Plasma Analysis Code (DynaPAC) [103]. The flight data and simulations indicated that the current collected had a power law dependence on the potential but a less than linear dependence on the plasma density. The measurements at low voltages, however, differed from the models as the latter predicted a threshold for current collection at -100 V which was not observed in the data [104].

The calculations just presented barely introduce the rich variety of low altitude plasma interactions now being studied. Consider the growing interest in electrodynamic tethers [105]–[108]. Multi-kilometer long thin conducting cables are now possible and have indeed been demonstrated (e.g., the TSS-1 Shuttle experiment [109]–[111]). One interest here is the use of these tethers to generate electricity. The basic principle is well known and contained in the Lorentz relationship

$$V = \vec{v}_s \times \vec{B} \cdot \vec{L} \tag{4}$$

where:

B magnetic field (vector);

L tether length (vector).

For a conducting object in low earth orbit, the $v \times B$ electric field varies from a low of about (0.1 V/m) at the equator to a maximum of (0.3 V/m) over the polar caps. As (4) states, the potential depends on the orientation of the tether relative to the $v \times B$ electric field vector. For a 10-km tether (easily possible with present technology), a potential difference of up to 3000 V is possible. A spacecraft can in principle draw power from this voltage drop but at the price of a loss in orbital altitude [112], [113]. By biasing a tether, it can be used for spacecraft propulsion and orbit reboost [114]–[117] as in the case of the Plasma Motor Generator [118] or the proposed Propulsive Small Expendable Deployer System (ProSEDS) experiment [114]. Tethers have even been suggested as possible power sources for jovian missions [119]. At Jupiter, however, the plasma corotates faster than the orbital velocity so that a vehicle can gain orbital altitude using a tether. Problems for tethers arise, however, in achieving the current flow necessary to utilize the energy as it is not clear that a sufficient ion current is possible without resorting to a plasma emitter or similar emission device [42], [107], [120], [121] to enhance positive ion current collection. Related issues are wave dissipation and radiation impedance associated with the passage of the tether [122]–[126]. Tethers are an on-going topic of research and debate.

Over the last decade, observations of surface charging on low earth orbit, polar spacecraft have mainly been concerned with the Defense Meteorological Satellite Program (DMSP) satellites. Papers [37], [127] have reported potentials ranging from a few hundreds of volts to over a kilovolt. It now appears that far from being a very rare event, moderate charging events (i.e., above the $\sim$400 V differential potentials normally believed to be the minimum necessary to cause surface arcing) are not uncommon for polar orbiting spacecraft. Discharge-induced anomalies, however, are believed to be rare. Recently, however, Anderson and Koons [128] have reported observing an operational anomaly on the DMSP F13 satellite; on May 5, 1995, the microwave imager experiment microprocessor locked-up. At the time of the event, the spacecraft frame potential was estimated at -460 V and surface potentials as high as -3 kV may have occurred in a 6 s period. Cooke [129] used the POLAR code to simulate the charging of the DMSP satellite at the time of the event. His results indicate that the highest potentials are only achieved by a few surfaces that have ion collection limited by their locations perhaps explaining in part the rarity of such events (though surface material choices may be a more likely cause). Even so, given adequate measurements of ionospheric and geomagnetic activity, it should be possible to determine the occurrence of such events in real time as in the case of geosynchronous charging.

Another area of low altitude charging interest is that associated with induced potentials due to biased surfaces such as solar arrays. In addition to arcing, biased solar arrays at low altitude have been observed to drive plasma effects (e.g., broadband fluctuations extending beyond 1 MHz) [130]. In a series of rocket and satellite experiments, the DoD and NASA have completed several interesting studies over the last decade into the effects of induced high potentials on solar arrays and of plasma beams on spacecraft potentials. Intended primarily to parameterize the ranges over which exposed high potential surfaces can be biased before arcing sets in and to demonstrate control of the discharge process, two series of experiments stand out. The first of these are the solar array experiments associated with the PASP Plus APEX satellite experiment [131], [132]. Launched into a 363 by 2550 km elliptical orbit on August 3, 1994 by a Pegasus rocket, this experiment consisted of a collection of several types of solar array cells. Ranging from solar concentrators to representative samples of the International Space Station arrays, the cells were biased over a range of voltages ($+500$ V) and their current collection and arcing characteristics were measured. In particular, the electron current collected by the so-called snap-over phenomena for positively biased solar arrays [69], [133], [134] was studied. Likewise, arcing at large negative potentials was also monitored [131], [132]. Pre-flight and post-flight simulations showed good agreement with the observations. Earlier ground experiments had found that the arc rate increases with voltage and plasma density and decreases with temperature [135], [136]. Other experiments [137]–[139] found an increase in arc rates with increasing voltage and plasma density. The PASP Plus results [136] demonstrated that arcing levels were indeed strongly dependent on bias voltage. Cell temperature was also verified as being critical with high arcing rates at low temperatures—a critical temperature above which no arcing occurred was observed. Finally, although radiation flux was found to have no effect on the arcing, the ion flux was seen to have an effect as expected [136].

The second low altitude charging experiments of interest are those associated with the Ballistic Missile Defense Organization's Space Power Experiment Aboard Rockets (SPEAR) Program [140]. In a series of three launches between 1987 and 1993, rockets were used to characterize the ability of a power system to maintain high voltages (upwards of 40 kV) in a dense ionospheric plasma ($\sim$200 to 300 km). These rocket flights were very successful in demonstrating the generation and control of multi-kilovolt potentials in dense, ionospheric plasma. Careful, ground-based studies permitted accurate modeling of the subsequent observations and detailed evaluations of a variety of techniques for controlling, measuring, and establishing high potentials in space without the need for heavy insulation relative to the plasma. Specifically, the SPEAR I [140], [141], launched in December 1987, was intended to investigate the interaction of a spacecraft with the ionospheric environment. It demonstrated that voltages >40 kV could be sustained in the space environment. SPEAR II was aimed at the actual operation of a high voltage system with minimum insulation (i.e., high voltage surfaces directly exposed to the LEO environment) and demonstrating technologies for utilizing the intrinsic insulating properties of the space vacuum as a form of high voltage insulation [142]. Although the rocket flight itself failed, the extensive ground tests performed at the NASA/Plum Brook large vacuum chamber on the SPEAR II systems demonstrated that a host of new high voltage technologies such as short circuit fault protection, high voltage capacitors, and rotating arc gap switches could operate in the space environment [140]. SPEAR III, successfully launched on March 15, 1993, combined these developments into a comprehensive test of methods for grounding high voltages in space (hollow cathodes, field emission, heated filaments, and neutral gas releases) [140], [143]–[145]. These results promise a new era in the utilization of high voltage systems in space.

A final issue to be considered is the application of these results to the International Space Station. The major issues associated with the International Space Station are its huge size and solar arrays. In addition to the preceding, various modeling efforts have attempted to address each of these features explicitly. For example, the floating potential and wake structure of the International Space Station and the likelihood of arcing have been extensively addressed by Hastings, Wang, and others [97], [146], [147]. Plasma contactors [148] in particular will play an important role in current collection and controlling floating potentials on the International Space Station's solar arrays and in limiting sputtering and arcing [147]. Other modeling programs specifically focusing on the International Space Station are briefly summarized by Katz *et al.* [149]. The practical aspects of effects of these plasma interactions (i.e., electromagnetic interference) are addressed by Murphy [150]. These papers, however, represent only a small portion of the plasma and charging studies that will ultimately come from our utilization of the International Space Station in the years ahead.

V. Conclusion

To summarize, the study and analysis of spacecraft charging over the last 20 years has demonstrated a growing maturity. Surface charging continues to be recognized as a serious operational threat to spacecraft and useful design guidelines are in place for its mitigation that were made possible in large part by the success of the SCATHA program. Internal charging has grown noticeably more important as a source of anomalies due to charging/arcing. With the flight of CRRES and its internal charging experiment, flight confirmation now exists of this phenomenon over the entire radiation belts. A formal internal charging design guideline was recently completed. Low altitude charging effects are slowly yielding to detailed computer analysis and experiment. Theory and evidence are converging on consistent models and techniques, successful conclusion of this process promises major advances in the utilization of the low altitude space environment. In particular, the use of tethers and of high voltage systems now appear possible if proper consideration is given to the details of the processes involved. For the future, there are, however, still many challenges. For example, the so-called "critical ionization velocity" phenomenon proposed by Alfven [151] remains an intriguing issue for low altitude plasma interactions [152]. The International Space Station itself promises to be a fertile laboratory setting for studying this and many other unusual plasma/charging interactions. Finally, there will be new areas that need to be investigated such as "dusty plasmas" [153] or the fields associated with truly large structures such as the proposed multi-kilometer solar sails. Even so, the last twenty years has seen significant and meaningful progress in an important scientific and engineering area of research, spacecraft charging.

The spacecraft charging phenomenon and its engineering implications for spacecraft design are having a resurgence. The very successful 6th Spacecraft Charging Technology Conference was held in November 1998 at the Air Force Research Lab, Hanscom Air Force Base, Massachusetts after a long hiatus between it and the 5th conference in 1989. A 7th Spacecraft Charging Technology Conference, sponsored by European Space Agency, will be held in Noordwijk, Holland in April 2001. In the United States, NASA's Marshall Space Flight Center (MSFC) has initiated a Space Environments Effects (SEE) program that seeks to provide a central focus to combine past successes with present and future efforts in spacecraft charging technology. These trends all point to a busy future for the field of spacecraft charging.

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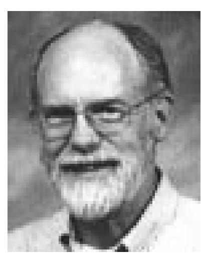

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